A|RE

IN THE UNITED STATES PATENT AND TRADEMARK OFFICE

In re Reissue Application of	)	Date: June 30, 1999
DAVID A. SPEAR ET AL.	)	Attorney Docket No.: 3600.100
Appln. No.: Not yet assigned	)	Examiner: Mark Sgantzos
Filed: Herewith	)	Group Art Unit: 3401
For: SWEPT TURBOMACHINERY BLADE	)	

"Express Mail" mailing label number EL 328767816 US
Date of Deposit June 30, 1999
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Karen Malatesta
KAREN MALATESTA

Assistant Commissioner for Patents
Box Patent Application
Washington, D.C. 20231

LETTER ACCOMPANYING REISSUE APPLICATION

Sir:

Pursuant to the practice under 37 C.F.R. § 1.53 and M.P.E.P. §§ 601.01, 1410, enclosed herewith for filing are the following papers constituting an application for reissue of United States Letters Patent No. 5,642,985, issued July 1, 1997, to David A. Spear, Bruce P. Biederman and John A. Orosa for SWEPT TURBOMACHINERY BLADE.

1. Mounted soft copy of the specification and new claims.
2. A copy of the patent drawings.
3. A soft copy of the printed original patent.
4. Proof of Executor's Authority to take action on behalf of David A. Spear, deceased.
5. Executed Reissue Declaration and Power of Attorney.

6. Assignee's Statement Under 37 C.F.R. § 3.73(b)

7. Assent of Assignee to Reissue Under 37 C.F.R. § 1.172.

8. Offer to Surrender Letters Patent.

9. The filing fee is computed as follows:

   a. Basic Filing Fee.................... $ 760.00

   b. Independent claims in excess of the number of independent claims in the original patent (7 X $78)...........$ 546.00

   c. Claims in excess of twenty and also in excess of the number of claims in the original patent (32 X $18)...........$ 576.00

Total: $ 1882.00

The Commissioner is hereby authorized to charge the above fee and any additional fees of any nature whatsoever under 37 C.F.R. § 1.16 and § 1.17 that may be required, or to credit any overpayment, to Deposit Account No. 21-0279. A duplicate of this letter is enclosed.

Pursuant to 37 C.F.R. § 1.174, please transfer the drawings from the patent file to the subject reissue application.

All correspondence should be directed to the following address:

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Respectfully submitted,

By___________________________
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For: SWEPT TURBOMACHINERY BLADE	)	Application to reissue U.S. Patent No. 5,642,985

Assistant Commissioner for Patents
Box Patent Application
Washington, D.C. 20231

ASSIGNEE'S STATEMENT UNDER 37 C.F.R. § 3.73(b)

Sir:

The undersigned, as representative of UNITED TECHNOLOGIES CORPORATION, certifies that, to the best of my knowledge and belief, title to United States Letters Patent No. 5,642,985 is in UNITED TECHNOLOGIES CORPORATION. This title is evidenced by an assignment recorded in the U.S. Patent and Trademark Office on November 17, 1995, at Reel 7772, Frame 0767.

UNITED TECHNOLOGIES CORPORATION

Date: 6/30/99

By: John Swiatocha
John Swiatocha
Assistant Secretary

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	)	
For: SWEPT TURBOMACHINERY BLADE	)	Application to reissue U.S. Patent No. 5,642,985

Assistant Commissioner for Patents
Box Patent Application
Washington, D.C. 20231

ASSENT OF ASSIGNEE TO
REISSUE UNDER 37 C.F.R. § 1.172

Sir:

The undersigned assignee of the entire interest in United States Letters Patent No. 5,642,985 hereby assents to the above-identified application to reissue such Letters Patent.

UNITED TECHNOLOGIES CORPORATION

Date: 6/30/99

By: John Swiatocha
John Swiatocha
Assistant Secretary

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Karen Malatesta

KAREN MALATESTA

Assistant Commissioner for Patents
Box Patent Application
Washington, D.C. 20231

OFFER TO SURRENDER LETTERS PATENT

Sir:

The undersigned applicants for the above-identified application for the reissue of Letters Patent No. 5,642,985, for an invention of a SWEPT TURBOMACHINERY BLADE, granted to them on July 1, 1997, the sole owner of which is UNITED TECHNOLOGIES CORPORATION, hereby offer to surrender said Letters Patent.

Date: 6/30/99

Dennis N. Kantor
Executor of the Estate of
David A. Spear, Deceased

Date: 6/28/99

Bruce P. Biederman

Date: 6-25-99

John A. Orosa

John A. Orosa

09/343659

FIDUCIARY'S PROBATE CERTIFICATE (PC-450) REV. 10/85

STATE OF CONNECTICUT
COURT OF PROBATE

Recorded:
Page: 1

Court of Probate, District of Manchester
District Number 077

95-0620

Estate of: Spear, David A.
aka David Alan Spear

Date of Certificate: 06/24/1999

Fiduciary's name and address:

Dennis N. Kantor, Box 280748, 330 Roberts Street, East Hartford, CT 06128-0748
Position of trust: Executor
Date of appointment: 11/28/1995

The undersigned hereby certifies that the fiduciary of the above *estate has accepted appointment; executed probate bond according to law or has been excused by will or by statute; and is legally authorized and qualified to act as such fiduciary on said estate; said appointment being unrevoked and in full force as of the above date of certificate.*

IN TESTIMONY WHEREOF, I have hereunto set my hand and affixed the seal of this court on the above date of certificate.

COURT SEAL

Mary C McNamara
Mary C. McNamara ~~XXXXX~~Judge/Ass't. Clerk

NOT VALID WITHOUT COURT OF PROBATE SEAL IMPRESSED

As used in this document, the word fiduciary includes the plural, where the context so requires.

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Karen Malatesta

KAREN MALATESTA

SWEPT TURBOMACHINERY BLADE

TECHNICAL FIELD

This invention relates to turbomachinery blades, and particularly to blades whose airfoils are swept to minimize the adverse effects of supersonic flow of a working medium over the airfoil surfaces.

BACKGROUND OF THE INVENTION

Gas turbine engines employ cascades of blades to exchange energy with a compressible working medium gas that flows axially through the engine. Each blade in the cascade has an attachment which engages a slot in a rotatable hub so that the blades extend radially outward from the hub. Each blade has a radially extending airfoil, and each airfoil cooperates with the airfoils of the neighboring blades to define a series of interblade flow passages through the cascade. The radially outer boundary of the flow passages is formed by a case which circumscribes the airfoil tips. The radially inner boundary of the passages is formed by abutting platforms which extend circumferentially from each blade.

During engine operation the hub, and therefore the blades attached thereto, rotate about a longitudinally extending rotational axis. The velocity of the working medium relative to the blades increases with increasing radius. Accordingly, it is not uncommon for the airfoil leading edges to be swept forward or swept back to mitigate the adverse aerodynamic effects associated with the compressibility of the working medium at high velocities.

One disadvantage of a swept blade results from pressure waves which extend along the span of each airfoil suction surface and reflect off the surrounding case. Because the airfoil is swept, both the incident waves and the reflected waves are oblique to the case. The reflected waves interact with the incident waves and coalesce into a planar aerodynamic shock which extends across the interblade flow channel between neighboring airfoils. These "endwall shocks" extend radially inward a limited distance from the case. In addition, the compressibility of the working medium causes a passage shock, which is unrelated to the above described endwall shock, to extend across the passage from the leading edge of each blade to the suction surface of the adjacent blade. As a result, the working medium gas flowing into the channels encounters multiple shocks and experiences unrecoverable losses in velocity and total pressure, both of which degrade the engine's efficiency. What is needed is a turbomachinery blade whose airfoil is swept to mitigate the effects of working medium compressibility while also avoiding the adverse influences of multiple shocks.

DISCLOSURE OF THE INVENTION

It is therefore an object of the invention to minimize the aerodynamic losses and efficiency degradation associated with endwall shocks by limiting the number of shocks in each interblade passage.

According to the invention, a blade for a blade cascade has an airfoil which is swept over at least a portion of its span, and the section of the airfoil radially coextensive with the endwall shock intercepts the endwall shock extending from the neighboring airfoil so that the endwall shock and the passage shock are coincident.

In one embodiment the axially forwardmost extremity of the airfoil's leading edge defines an inner transition point located at an inner transition radius radially inward of the

airfoil tip. An outer transition point is located at an outer transition radius radially intermediate the inner transition radius and the airfoil tip. The outer transition radius and the tip bound a blade tip region while the inner and outer transition radii bound an intermediate region. The leading edge is swept at a first sweep angle in the intermediate region and is swept at a second sweep angle over at least a portion of the tip region. The first sweep angle is generally nondecreasing with increasing radius and the second sweep angle is generally non-increasing with increasing radius.

The invention has the advantage of limiting the number of shocks in each interblade passage so that engine efficiency is maximized.

BRIEF DESCRIPTION OF THE DRAWINGS

FIG. 1 is a cross sectional side elevation of the fan section of a gas turbine engine showing a swept back fan blade according to the present invention.

FIG. 2 is an enlarged view of the blade of FIG. 1 including an alternative leading edge profile shown by dotted lines and a prior art blade shown in phantom.

FIG. 3 is a developed view taken along the line 3—3 of FIG. 2 illustrating the tips of four blades of the present invention along with four prior art blades shown in phantom.

FIG. 4 is a schematic perspective view of an airfoil fragment illustrating the definition of sweep angle.

FIG. 5 is a developed view similar to FIG. 3 illustrating an alternative embodiment of the invention and showing prior art blades in phantom.

FIG. 6 is a cross sectional side elevation of the fan section of a gas turbine engine showing a forward swept fan blade according to the present invention and showing a prior art fan blade in phantom.

FIG. 7 is a developed view taken along the line 7—7 of FIG. 6 illustrating the tips of four blades of the present invention along with four prior art blades shown in phantom.

BEST MODE FOR CARRYING OUT THE INVENTION

Referring to FIGS. 1–3, the forward end of a gas turbine engine includes a fan section **10** having a cascade of fan blades **12**. Each blade has an attachment **14** for attaching the blade to a disk or hub **16** which is rotatable about a longitudinally extending rotational axis **18**. Each blade also has a circumferentially extending platform **20** radially outward of the attachment. When installed in an engine, the platforms of neighboring blades in the cascade abut each other to form the cascade's inner flowpath boundary. An airfoil **22** extending radially outward from each platform has a root **24**, a tip **26**, a leading edge **28**, a trailing edge **30**, a pressure surface **32** and a suction surface **34**. The axially forwardmost extremity of the leading edge defines an inner transition point **40** at an inner transition radius r_t-inner, radially inward of the tip. The blade cascade is circumscribed by a case **42** which forms the cascade's outer flowpath boundary. The case includes a rubstrip **46** which partially abrades away in the event that a rotating blade contacts the case during engine operation. A working medium fluid such as air **48** is pressurized as it flows axially through interblade passages **50** between neighboring airfoils.

The hub **16** is attached to a shaft **52**. During engine operation, a turbine (not shown) rotates the shaft, and therefore the hub and the blades, about the axis **18** in direction R. Each blade, therefore, has a leading neighbor

which precedes it and a trailing neighbor which follows it during rotation of the blades about the rotational axis.

The axial velocity V_x (FIG. 3) of the working medium is substantially constant across the radius of the flowpath. However the linear velocity U of a rotating airfoil increases with increasing radius. Accordingly, the relative velocity V_r of the working medium at the airfoil leading edge increases with increasing radius, and at high enough rotational speeds, the airfoil experiences supersonic working medium flow velocities in the vicinity of its tip. Supersonic flow over an airfoil, while beneficial for maximizing the pressurization of the working medium, has the undesirable effect of reducing fan efficiency by introducing losses in the working medium's velocity and total pressure. Therefore, it is typical to sweep the airfoil's leading edge over at least a portion of the blade span so that the working medium velocity component in the chordwise direction (perpendicular to the leading edge) is subsonic. Since the relative velocity V_r increases with increasing radius, the sweep angle typically increases with increasing radius as well. As shown in FIG. 4, the sweep angle σ at any arbitrary radius is the acute angle between a line **54** tangent to the leading edge **28** of the airfoil **22** and a plane **56** perpendicular to the relative velocity vector V_r. The sweep angle is measured in plane **58** which contains both the relative velocity vector and the tangent line and is perpendicular to plane **56**. In conformance with this definition sweep angles σ_1 and σ_2, referred to hereinafter and illustrated in FIGS. **2**, **3** and **6** are shown as projections of the actual sweep angle onto the plane of the illustrations.

Sweeping the blade leading edge, while useful for minimizing the adverse effects of supersonic working medium velocity, has the undesirable side effect of creating an endwall reflection shock. The flow of the working medium over the blade suction surface generates pressure waves **60** (shown only in FIG. **1**) which extend along the span of the blade and reflect off the case. The reflected waves **62** and the incident waves **60** coalesce in the vicinity of the case to form an endwall shock **64** across each interblade passage. The endwall shock extends radially inward a limited distance, d, from the case. As best seen in the prior art (phantom) illustration of FIG. **3**, each endwall shock is also oblique to a plane **67** perpendicular to the rotational axis so that the shock extends axially and circumferentially. In principle, an endwall shock can extend across multiple interblade passages and affect the working medium entering those passages. In practice, expansion waves (as illustrated by the representative waves **68**) propagate axially forward from each airfoil and weaken the endwall shock from the airfoil's leading neighbor so that each endwall shock usually affects only the passage where the endwall shock originated. In addition, the supersonic character of the flow causes passage shocks **66** to extend across the passages. The passage shocks, which are unrelated to endwall reflections, extend from the leading edge of each blade to the suction surface of the blade's leading neighbor. Thus, the working medium is subjected to the aerodynamic losses of multiple shocks with a corresponding degradation of engine efficiency.

The endwall shock can be eliminated by making the case wall perpendicular to the incident expansion waves so that the incident waves coincide with their reflections. However other design considerations, such as constraints on the flowpath area and limitations on the case construction, may make this option unattractive or unavailable. In circumstances where the endwall shock cannot be eliminated, it is desirable for the endwall shock to coincide with the passage shock since the aerodynamic penalty of coincident shocks is less than that of multiple individual shocks.

According to the present invention, coincidence of the endwall shock and the passage shock is achieved by uniquely shaping the airfoil so that the airfoil intercepts the endwall shock extending from the airfoil's leading neighbor and results in coincidence between the endwall shock and the passage shock.

A swept back airfoil according to the present invention has a leading edge **28**, a trailing edge **30**, a root **24** and a tip **26** located at a tip radius r_{tip}. An inner transition point **40** located at an inner transition radius r_r-inner is the axially forwardmost point on the leading edge. The leading edge of the airfoil is swept back by a radially varying first sweep angle σ_1 in an intermediate region **70** of the airfoil (in FIG. **2** plane **56** appears as the line defined by the plane's intersection with the plane of the illustration and in FIG. **3** the tangent line **54** appears as the point where the tangent line penetrates the plane of the Figure). The intermediate region **70** is the region radially bounded by the inner transition radius r_r-inner and the outer transition radius r_r-outer. The first sweep angle, as is customary in the art, is nondecreasing with increasing radius, i.e. the sweep angle increases, or at least does not decrease, with increasing radius.

The leading edge **28** of the airfoil is also swept back by a radially varying second sweep angle σ_2 in a tip region **74** of the airfoil. The tip region is radially bounded by the outer transition radius r_r-outer and a tip radius r_{tip}. The second sweep angle is nonincreasing (decreases, or at least does not increase) with increasing radius. This is in sharp contrast to the prior art airfoil **22'** whose sweep angle increases with increasing radius radially outward of the inner transition radius.

The beneficial effect of the invention is appreciated primarily by reference to FIG. **3** which compares the invention (and the associated endwall and passage shocks) to a prior art blade (and its associated shocks) shown in phantom. Referring first to the prior art illustration in phantom, the endwall shock **64** originates as a result of the pressure waves **60** (FIG. **1**) extending along the suction surface of each blade. Each endwall shock is oblique to a plane **67** perpendicular to the rotational axis, and extends across the inter-blade passage of origin. The passage shock **66** also extends across the flow passage from the leading edge of a blade to the suction surface of the blade's leading neighbor. The working medium entering the passages is therefore adversely influenced by multiple shocks. By contrast, the nonincreasing character of the second sweep angle of a swept back airfoil **22** according to the invention causes a portion of the airfoil leading edge to be far enough forward (upstream) in the working medium flow that the section of the airfoil radially coextensive with the endwall shock extending from the airfoil's leading neighbor intercepts the endwall shock **64** (the unique sweep of the airfoil does not appreciably affect the location or orientation of the endwall shock; the phantom endwall shock associated with the prior art blade is illustrated slightly upstream of the endwall shock for the airfoil of the invention for illustrative clarity). In addition, the passage shock **66** (which remains attached to the airfoil leading edge and therefore is translated forward along with the leading edge) is brought into coincidence with the endwall shock so that the working medium does not encounter multiple shocks.

The embodiment of FIGS. **2** and **3** illustrates a blade whose leading edge, in comparison to the leading edge of a conventional blade, has been translated axially forward parallel to the rotational axis (the corresponding translation of the trailing edge is an illustrative convenience—the

location of the trailing edge is not embraced by the invention). However the invention contemplates any blade whose airfoil intercepts the endwall shock to bring the passage shock into coincidence with the endwall shock. For example, FIG. 5 illustrates an embodiment where a section of the tip region is displaced circumferentially (relative to the prior art blade) so that the blade intercepts the endwall shock **64** and brings it into coincidence with the passage shock **66**. As with the embodiment of FIG. 3, the displaced section extends radially inward far enough to intercept the endwall shock over its entire radial extent and brings it into coincidence with the passage shock **66**. This embodiment functions as effectively as the embodiment of FIG. 3 in terms of bringing the passage shock into coincidence with the endwall shock. However it suffers from the disadvantage that the airfoil tip is curled in the direction of rotation R. In the event that the blade tip contacts the rubstrip **46** during engine operation, the curled blade tip will gouge rather than abrade the rubstrip necessitating its replacement. Other alternative embodiments may also suffer from this or other disadvantages.

The invention's beneficial effects also apply to a blade having a forward swept airfoil. Referring to FIG. **6** and **7**, a forward swept airfoil **122** according to the present invention has a leading edge **128**, a trailing edge **130**, a root **124** and a tip **126** located at a tip radius r_{tip}. An inner transition point **140** located at an inner transition radius r_t-inner is the axially aftmost point on the leading edge. The leading edge of the airfoil is swept forward by a radially varying first sweep angle σ_1 in an intermediate region **70** of the airfoil. The intermediate region is radially bounded by the inner transition radius r_t-inner and the outer transition radius r_t-outer. The first sweep angle [τ_1] σ_1 is nondecreasing with increasing radius, i.e. the sweep angle increases, or at least does not decrease, with increasing radius.

The leading edge ~~128~~ of the airfoil is also swept forward by a radially varying second sweep angle σ_2 in a tip region **74** of the airfoil. The tip region is radially bounded by the outer transition radius r_t-outer and the tip radius r_{tip}. The second sweep angle is nonincreasing (decreases, or at least does not increase) with increasing radius. This is in sharp contrast to the prior art airfoil **122'** whose sweep angle increases with increasing radius radially outward of the inner transition radius.

In the forward swept embodiment of the invention, as in the swept back embodiment, the nonincreasing sweep angle σ_2 in the tip region **74** causes the endwall shock **64** to be coincident with the passage shock **66** for reducing the aerodynamic losses as discussed previously. This is in contrast to the prior art blade, shown in phantom where the endwall shock and the passage shock are distinct and therefore impose multiple aerodynamic losses on the working medium.

In the swept back embodiment of FIG. **2**, the inner transition point is the axially forwardmost point on the leading edge. The leading edge is swept back at radii greater than the inner transition radius. The character of the leading edge sweep inward of the inner transition radius is not embraced by the invention. In the forward swept embodiment of FIG. **6**, the inner transition point is the axially aftmost point on the leading edge. The leading edge is swept forward at radii greater than the inner transition radius. As with the swept back embodiment, the character of the leading edge sweep inward of the inner transition radius is not embraced by the invention. In both the forward swept

and back swept embodiments. the inner transition point is illustrated as being radially outward of the airfoil root. However the invention also comprehends a blade whose inner transition point (axially forwardmost point for the swept back embodiment and axially aftmost point for the forward swept embodiment) is radially coincident with the leading edge of the root. This is shown. for example. by the dotted leading edge 28" of FIG. 2.

The invention has been presented in the context of a fan blade for a gas turbine engine. however. the invention's applicability extends to any turbomachinery airfoil wherein flow passages between neighboring airfoils are subjected to multiple shocks.

We claim:

1. A turbomachinery blade for a turbine engine having a cascade of blades rotatable about a rotational axis so that each blade in the cascade has a leading neighbor and a trailing neighbor. and each blade cooperates with its neighbors to define flow passages for a working medium gas, the blade cascade being circumscribed by a case and under some operational conditions an endwall shock extends a limited distance radially inward from the case and also extends axially and circumferentially across the flow passages, and a passage shock also extends across the flow passages, the turbomachinery blade including an airfoil having a leading edge, a trailing edge. a root, a tip and an inner transition point located at an inner transition radius radially inward of the tip. the blade characterized in that at least a portion of the leading edge radially outward of the inner transition point is swept and a section of the airfoil radially coextensive with the endwall shock extending from the leading neighbor intercepts the endwall shock so that the endwall shock and the passage shock are coincident.

2. A turbomachinery blade for a turbine engine having a cascade of blades rotatable about a rotational axis so that each blade in the cascade has a leading neighbor and a trailing neighbor, and each blade cooperates with its neighbors to define flow passages for a working medium gas, the blade cascade being circumscribed by a case and under some operational conditions an endwall shock extends a limited distance radially inward from the case and also extends axially and circumferentially across the flow passages and a passage shock also extends across the flow passages. the turbomachinery blade including an airfoil having a leading edge, a trailing edge, a root, a tip located at a tip radius, an inner transition point located at an inner transition radius radially inward of the tip. and an outer transition point at an outer transition radius radially intermediate the inner transition radius and the tip radius, the blade having a tip region bounded by the outer transition radius and the tip radius. and an intermediate region bounded by the inner transition radius and the outer transition radius. the blade characterized in that the leading edge is swept in the intermediate region at a first sweep angle which is generally nondecreasing with increasing radius. and the leading edge is swept over at least a portion of the tip region at a second sweep angle which is generally nonincreasing with increasing radius so that the section of the airfoil radially coextensive with the endwall shock extending from the leading neighbor intercepts the endwall shock so that the endwall shock and the passage shock are coincident.

3. The turbomachinery blade of claim 1 or 2 characterized in that the inner transition radius is coincident with the root at the leading edge of the blade.

4. A turbomachinery blade for a gas turbine engine fan comprising a plurality of blades mounted for rotation about a fan axis with neighboring blades forming passages for a working medium gas, wherein:

the blade has a configuration enabling the fan to rotate at speeds providing supersonic flow velocities in at least a portion of each passage causing the formation of a shock in the gas adjacent an inner wall of a case forming an outer boundary for the working medium gas flowing through the passages;

the blade has a leading edge with an intermediate region and a tip region outward of the intermediate region and extending to a tip end of the blade, the intermediate region being swept rearward at a sweep angle that does not decrease; and

the tip region is translated forward to provide a sweep angle that causes the blade to intercept the shock.

5. The turbomachinery blade of claim 4, wherein the tip region begins at an outward boundary of the intermediate region and throughout the tip region the sweep angle is less than the sweep angle at the outward boundary of the intermediate region.

6. The turbomachinery blade of claim 5, wherein the sweep angle decreases throughout the tip region.

7. The turbomachinery blade of claim 6, wherein the sweep angle increases throughout the intermediate region.

8. The turbomachinery blade of any one of claims 4 to 7, wherein an inward boundary of the intermediate region is coincident with a root end of the blade.

9. The turbomachinery blade of any one of claims 4 to 7, wherein the leading edge of the blade has an inner region beginning at a root end of the blade and extending to an inward boundary of the intermediate region, the inner region being swept forward.

10. A blade for a gas turbine engine fan comprising a plurality of blades mounted for rotation within a case circumscribing the blades and forming an outer boundary for a working medium gas flowing through passages formed by neighboring blades, wherein:

the blade has a configuration enabling the fan to rotate at speeds providing supersonic flow velocities in at least a portion of each passage;

the blade has a leading edge with an intermediate region and a tip region beginning at an outward boundary of the intermediate region and extending to a tip end of the blade, the intermediate region having a sweep angle that does not decrease from the beginning to the outward boundary of the intermediate region; and

throughout the tip region the sweep angle is less than the sweep angle at the outward boundary of the intermediate region.

11. The blade of claim 10, wherein the intermediate region is swept rearward and the tip region is translated forward.

12. The blade of claim 10, wherein the intermediate region is swept forward and the tip region is translated rearward.

13. The blade of claim 10, wherein the tip region sweep angle decreases throughout the tip region.

14. The blade of claim 13, wherein the intermediate region sweep angle increases throughout the intermediate region.

15. The blade of any one of claims 10 to 14, wherein an inward boundary of the intermediate region is coincident with a root end of the blade.

16. The blade of claim 10, wherein:

the intermediate region is swept rearward and the tip region is translated forward; and

the leading edge of the blade has an inner region beginning at a root end of the blade and extending to an inward boundary of the intermediate region, the inner region being swept forward.

17. The blade of claim 16, wherein:

the intermediate region sweep angle increases throughout the intermediate region; and

the tip region sweep angle decreases throughout the tip region.

18. A blade for a gas turbine engine fan comprising a plurality of blades mounted for rotation within a case circumscribing the blades and forming an outer boundary for a working medium gas flowing through passages formed by neighboring blades, wherein:

the blade has a configuration enabling the fan to rotate at speeds providing supersonic flow velocities in at least a portion of each passage;

the blade has a leading edge with an intermediate region and a tip region beginning at an outward boundary of the intermediate region and extending to a tip end of the blade, the intermediate region being swept rearward at a sweep angle that does not decrease from the beginning to the outward boundary of the intermediate region; and

the tip region is translated forward from the outward boundary of the rearwardly swept intermediate region.

19. The blade of claim 18, wherein the tip region maintains a rearward sweep throughout the tip region.

20. Turbomachinery for a gas turbine engine, comprising a plurality of blades mounted for rotation within a case circumscribing the blades and forming an outer boundary for a working medium gas flowing through passages formed by neighboring blades, wherein:

each blade has a configuration enabling the turbomachinery to rotate at speeds providing supersonic working medium gas velocities at least in the vicinity of the passages proximate to the case;

each blade has a leading edge with a swept intermediate region and a swept tip region beginning at an outward boundary of the intermediate region and extending to a tip end of the blade, the intermediate region of each blade having a sweep angle that does not decrease from the beginning to the outward boundary of the intermediate region; and

throughout the tip region the sweep angle of each blade is less than the sweep angle at the outward boundary of the intermediate region.

21. The turbomachinery of claim 20, wherein the intermediate region of each blade is swept rearward and the tip region is translated forward.

22. The turbomachinery of claim 21, wherein:

the intermediate region sweep angle of each blade increases throughout the intermediate region; and

the tip region sweep angle of each blade decreases throughout the tip region.

23. The turbomachinery of claim 22, wherein the leading edge of each blade has an inner region beginning at a root end of the blade and extending to an inward boundary of the intermediate region, the inner region being swept forward.

24. The turbomachinery of claim 20, wherein the intermediate region of each blade is swept forward and the tip region is translated rearward.

25. The turbomachinery of claim 24, wherein the tip region sweep angle of each blade decreases throughout the tip region.

26. The turbomachinery of claim 25, wherein the intermediate region sweep angle of each blade increases throughout the intermediate region.

27. A gas turbine engine fan comprising a plurality of identical blades, each blade being mounted for rotation within a case circumscribing the blades and having an inner wall forming an outer boundary for a working medium gas flowing through passages formed by neighboring blades, wherein:

each blade has a configuration enabling the fan to rotate at speeds providing supersonic working medium gas velocities in the vicinity of the passages proximate to the case;

each blade has a leading edge with an inner region, an intermediate region and a tip region, the inner region beginning at a root end of

the blade and extending to an inward boundary of the intermediate region, and the tip region extending from an outward boundary of the intermediate region to a tip end of the blade; and

the inner region is swept forward, the intermediate region is swept rearward at a sweep angle that does not decrease, and the tip region is translated forward from the outward boundary of the intermediate region.

28. The gas turbine engine fan of claim 27, wherein the tip region maintains a rearward sweep throughout the tip region.

29. The gas turbine engine fan of claim 27, wherein:

the intermediate region sweep angle of each blade increases throughout the intermediate region; and

the tip region of each blade is swept at a sweep angle that decreases throughout the tip region.

30. A blade for a gas turbine engine rotatable within a case at speeds providing supersonic flow over at least a portion of the blade, wherein the blade leading edge has a rear swept middle region ending at a tip region that is translated forward from the end of the middle region.

31. The blade of claim 30, wherein the tip region maintains a rearward sweep throughout the tip region.

32. The blade of claim 30, wherein the leading edge has a forward swept inner region.

33. The blade of claim 32, wherein the sweep angle of the middle region increases throughout the middle region.

34. The blade of claim 33, wherein throughout the tip region the sweep angle is less than the sweep angle at the end of the middle region.

35. The blade of claim 34, wherein the sweep angle of the tip region decreases from the end of the middle region to a tip end of the blade.

36. A blade for a gas turbine engine rotatable within a case at speeds providing supersonic flow over at least a portion of the blade, wherein the blade leading edge has a forward swept middle region ending at a tip region that is translated rearward from the end of the middle region.

37. The blade of claim 36, wherein the tip region maintains a forward sweep throughout the tip region.

38. The blade of claim 36, wherein the leading edge has a rear swept inner region.

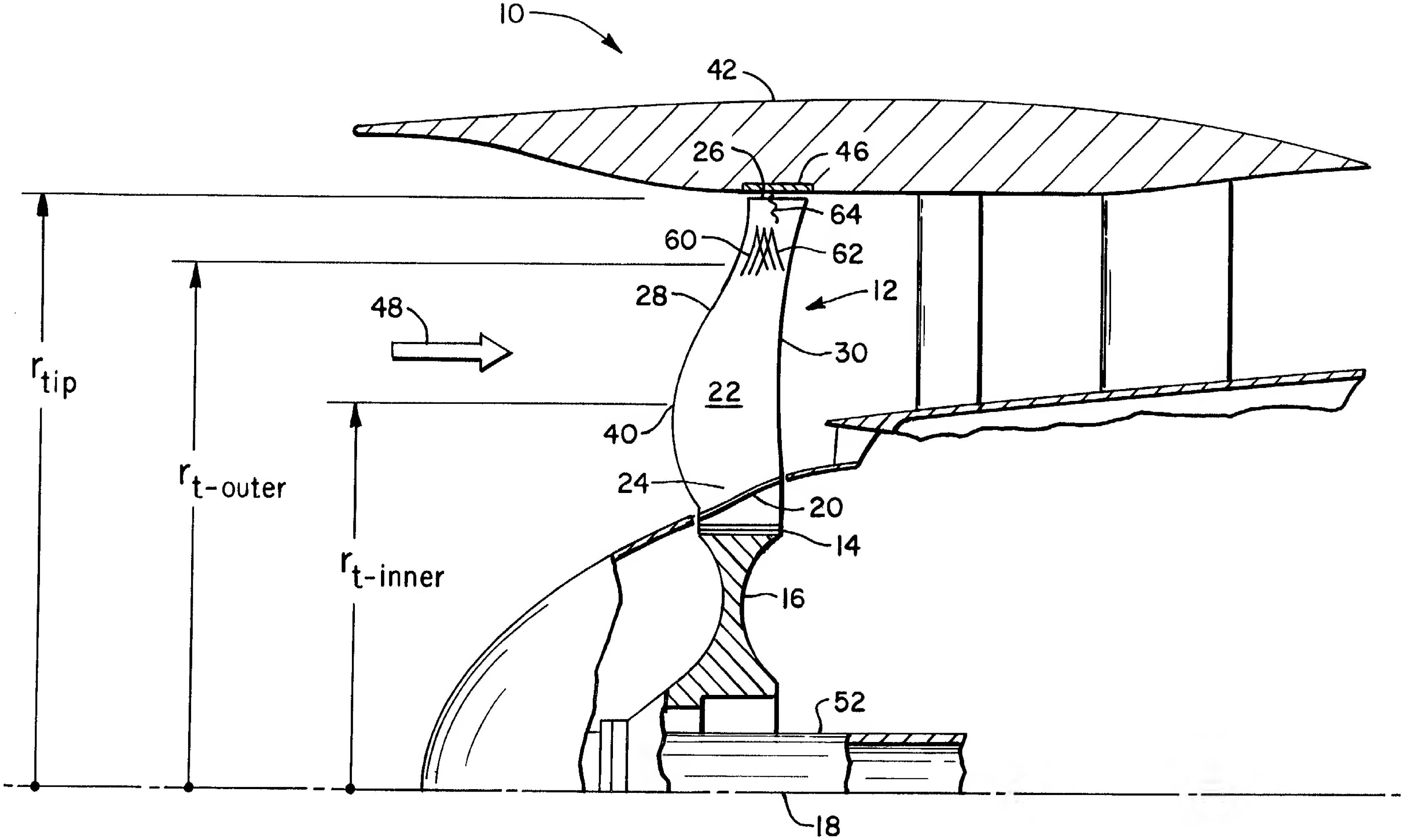
39. The blade of claim 38, wherein the sweep angle of the middle region increases throughout the middle region.

40. The blade of claim 39, wherein throughout the tip region the sweep angle is less than the sweep angle at the end of the middle region.

41. The blade of claim 40, wherein the sweep angle of the tip region decreases from the end of the middle region to a tip end of the blade.

ABSTRACT

A swept turbomachinery blade for use in a cascade of such blades is disclosed. The blade (12) has an airfoil (22) uniquely swept so that an endwall shock (64) of limited radial extent and a passage shock (66) are coincident and a working medium (48) flowing through interblade passages (50) is subjected to a single coincident shock rather than the individual shocks. In one embodiment of the invention the forwardmost extremity of the airfoil defines an inner transition point (40) located at an inner transition radius r_t-inner. The sweep angle of the airfoil is nondecreasing with increasing radius from the inner transition radius to an outer transition radius $r_{t\text{-}outer}$, radially inward of the airfoil tip (26), and is nonincreasing with increasing radius between the outer transition radius and the airfoil tip.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12
48
30
22
40
24
20
14
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52
18
r_{tip}
$r_{t\text{-}outer}$
$r_{t\text{-}inner}$

FIG. 1

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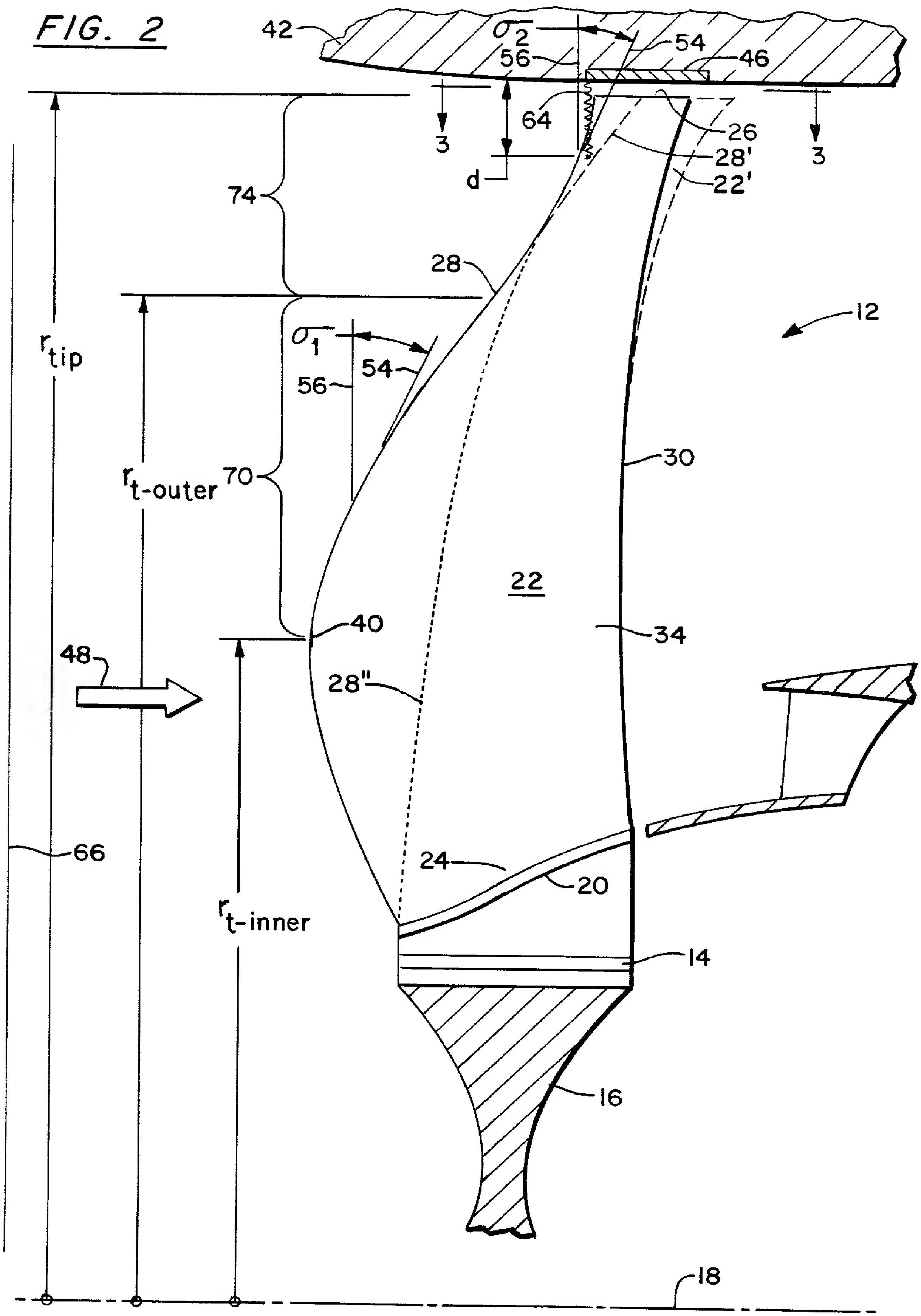
FIG. 2



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56
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d
22'
74
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r_{tip}
σ_1
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56
30
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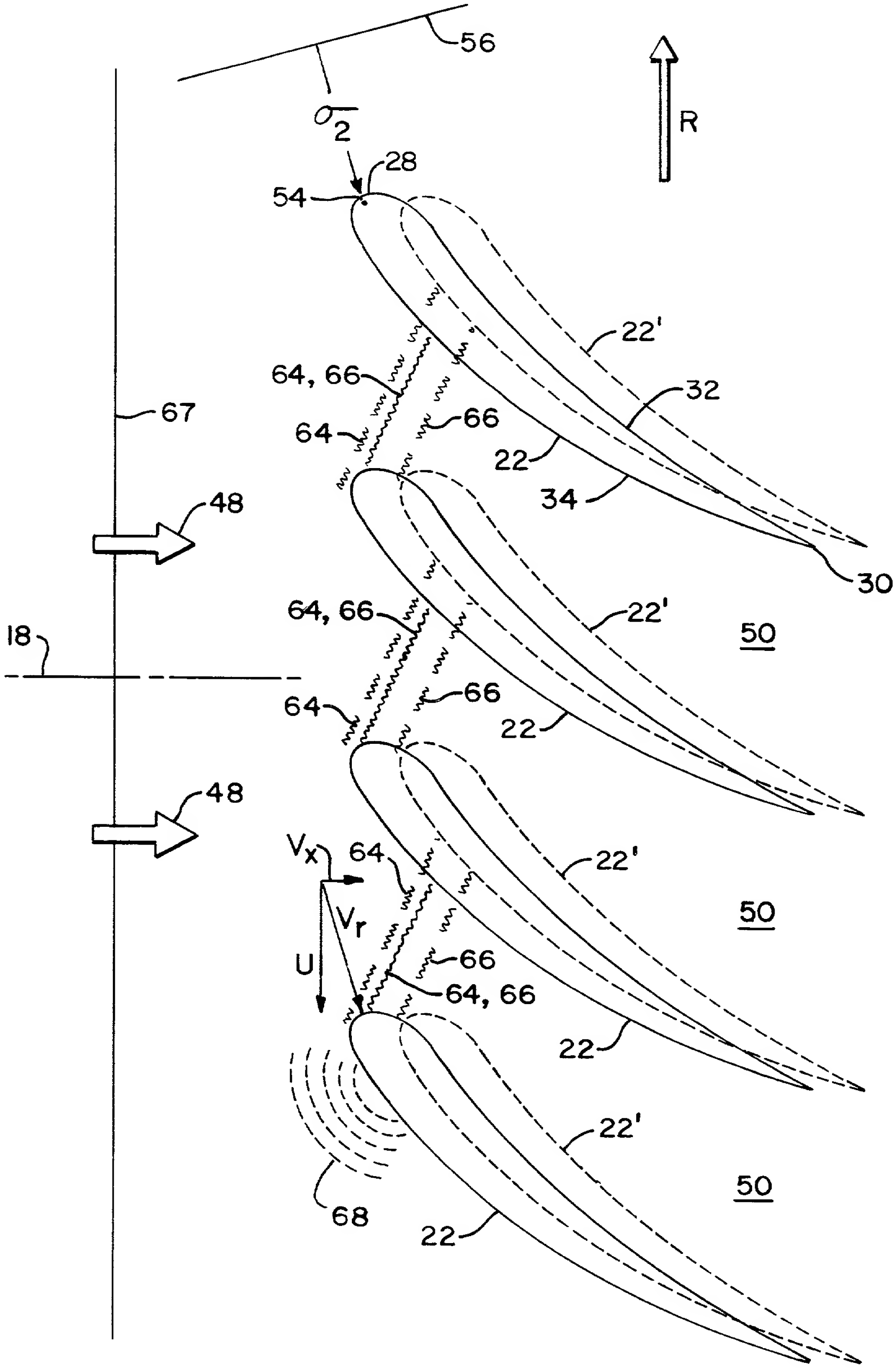
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68
22

FIG. 3

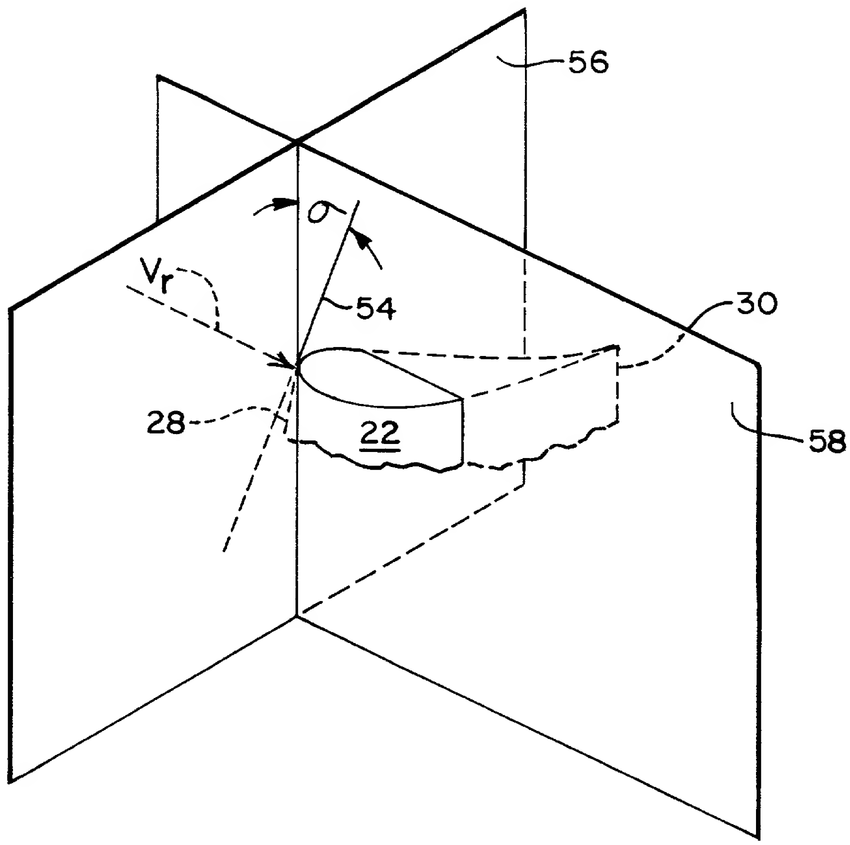
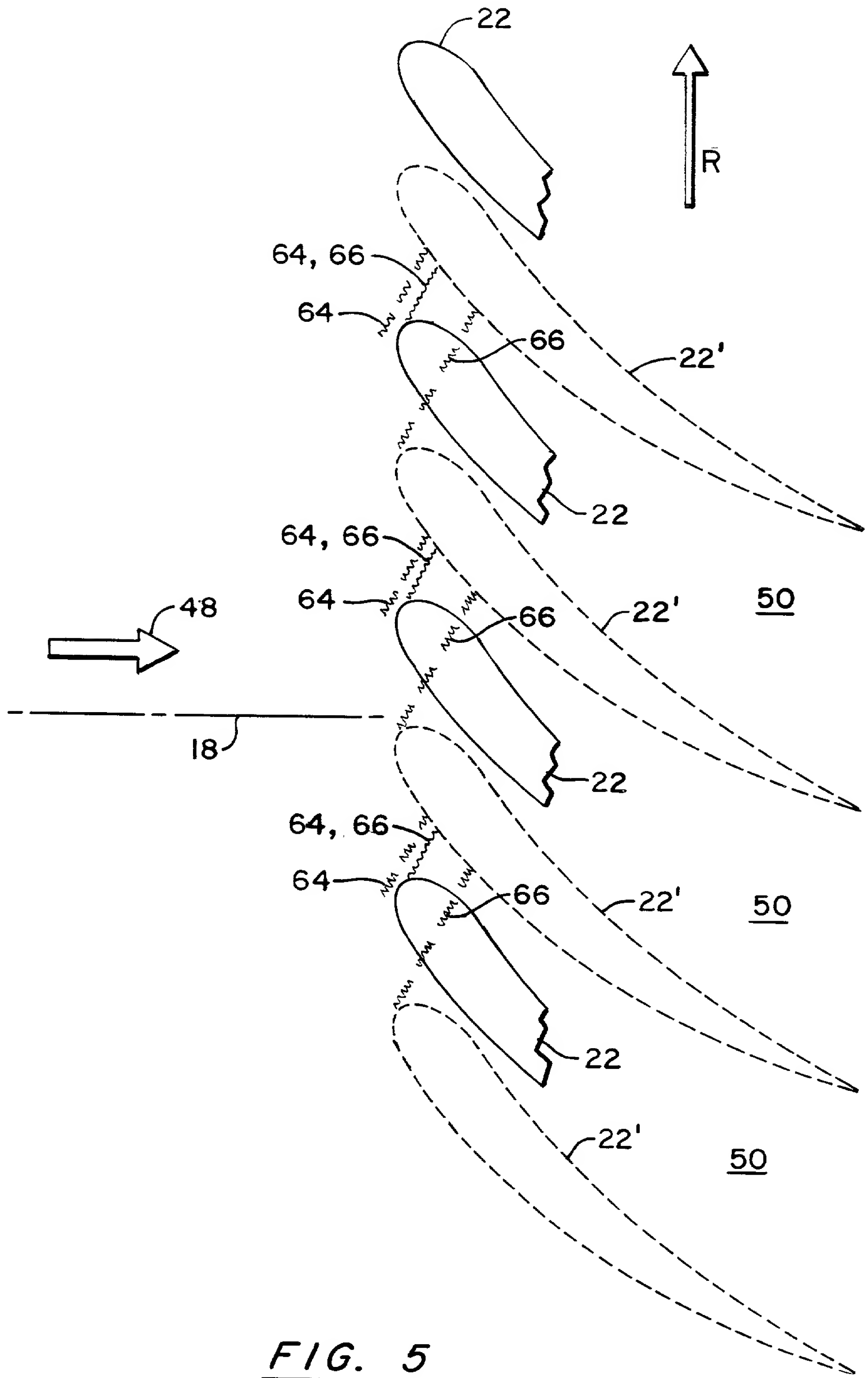


FIG. 4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22'
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22'
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22
22'
50

FIG. 5

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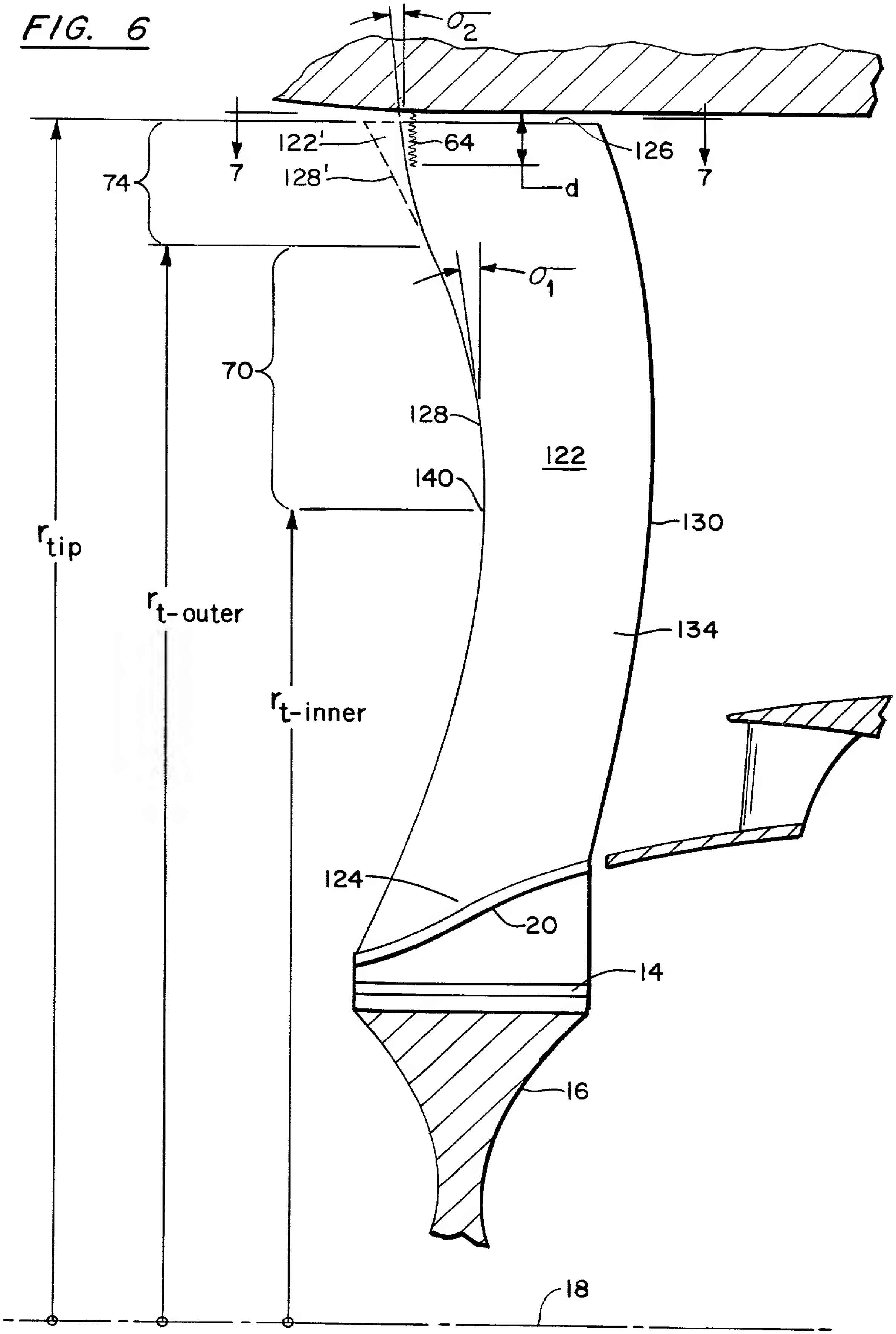
FIG. 6



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$r_{t-inner}$
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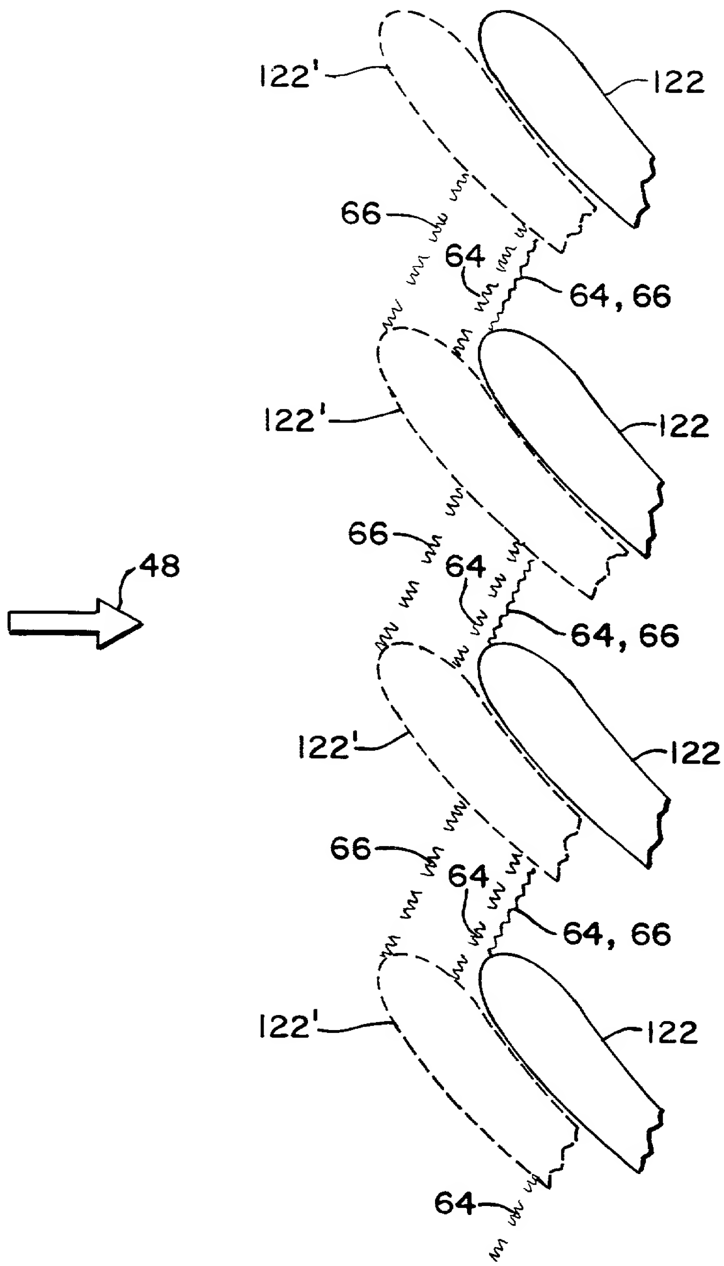
18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122'
122
64

FIG. 7

IN THE UNITED STATES PATENT AND TRADEMARK OFFICE

In re Reissue Application of DAVID A. SPEAR ET AL. Appln. No.: Not yet assigned Filed: Herewith For: SWEPT TURBOMACHINERY BLADE	)	Attorney Docket No.: 3600.100 Examiner: Mark Sgantzos Group Art Unit: 3401 Application to reissue U.S. Patent No. 5,642,985

"Express Mail" mailing label number EL328767816US
Date of Deposit Aug. 30, 1999
I hereby certify that this paper or fee is being deposited with the United States Postal Service "Express Mail Post Office to Addressee." service under 37 CFR 1.10 on the date indicated above and is addressed to the Commissioner of Patents and Trademarks Washington DC 20231
Karen Malatesta
KAREN MALATESTA

Assistant Commissioner for Patents
Box Patent Application
Washington, D.C. 20231

REISSUE DECLARATION AND POWER OF ATTORNEY

Sir:

We, DAVID A. SPEAR, who was a United States citizen residing at Manchester, Connecticut, at the time of his death on October 22, 1995, BRUCE P. BIEDERMAN, a United States citizen residing at West Hartford, Connecticut, and JOHN A. OROSA, a United States citizen residing at Palm Beach Gardens, Florida, hereby declare and say that:

1. We believe that we are the original, first and joint inventors of the subject matter which is claimed in the subject reissue application and for which a reissue patent is sought on the invention entitled SWEPT TURBOMACHINERY BLADE, the specification of which is filed herewith.

2. We have reviewed and understand the contents of the reissue application, including the claims.

3. We acknowledge our duty to disclose to the U.S. Patent and Trademark Office all information known to be material to patentability as defined in 37 C.F.R. § 1.56.

4. We believe that the original above-identified U.S. Patent No. 5,642,985 is partly inoperative by reason of us having claimed less than we had the right to claim in that patent. Specifically, we believe that we were entitled to claims to at least the following subject matter:

A turbomachinery blade for a gas turbine engine fan comprising a plurality of blades mounted for rotation about a fan axis with neighboring blades forming passages for a working medium gas, wherein:

the blade has a configuration enabling the fan to rotate at speeds providing supersonic flow velocities in at least a portion of each passage causing the formation of a shock in the gas adjacent an inner wall of a case forming an outer boundary for the working medium gas flowing through the passages;

the blade has a leading edge with an intermediate region and a tip region outward of the intermediate region and extending to a tip end of the blade, the intermediate region being swept rearward at a sweep angle that does not decrease; and

the tip region is translated forward to provide a sweep angle that causes the blade to intercept the shock.

A blade for a gas turbine engine fan comprising a plurality of blades mounted for rotation within a case circumscribing the blades and forming an outer boundary for a working medium gas flowing through passages formed by neighboring blades, wherein:

the blade has a configuration enabling the fan to rotate at speeds providing supersonic flow velocities in at least a portion of each passage;

the blade has a leading edge with an intermediate region and a tip region beginning at an outward boundary of the intermediate region and extending to a tip end of the blade, the intermediate region having a sweep angle that does not decrease from the beginning to the outward boundary of the intermediate region; and

throughout the tip region the sweep angle is less than the sweep angle at the outward boundary of the intermediate region.

A blade for a gas turbine engine fan comprising a plurality of blades mounted for rotation within a case circumscribing the blades and forming an outer boundary for a working medium gas flowing through passages formed by neighboring blades, wherein:

the blade has a configuration enabling the fan to rotate at speeds providing supersonic flow velocities in at least a portion of each passage;

the blade has a leading edge with an intermediate region and a tip region beginning at an outward boundary of the intermediate region and extending to a tip end of the blade, the intermediate region being swept rearward at a sweep angle that does not decrease from the beginning to the outward boundary of the intermediate region; and

the tip region is translated forward from the outward boundary of the rearwardly swept intermediate region.

Turbomachinery for a gas turbine engine, comprising a plurality of blades mounted for rotation within a case circumscribing the blades and forming an outer boundary for a working medium gas flowing through passages formed by neighboring blades, wherein:

each blade has a configuration enabling the turbomachinery to rotate at speeds providing supersonic working medium gas velocities at least in the vicinity of the passages proximate to the case;

each blade has a leading edge with a swept intermediate region and a swept tip region beginning at an outward boundary of the intermediate region and extending to a tip end of the blade, the intermediate region of each blade having a sweep angle that does not decrease from the beginning to the outward boundary of the intermediate region; and

throughout the tip region the sweep angle of each blade is less than the sweep angle at the outward boundary of the intermediate region.

A gas turbine engine fan comprising a plurality of identical blades, each blade being mounted for rotation within a case circumscribing the blades and having an inner wall forming an outer boundary for a working medium gas flowing through passages formed by neighboring blades, wherein:

each blade has a configuration enabling the fan to rotate at speeds providing supersonic working medium gas velocities in the vicinity of the passages proximate to the case;

each blade has a leading edge with an inner region, an intermediate region and a tip region, the inner region beginning at a root end of the blade and extending to an inward boundary of the intermediate region, and the tip region extending from an outward boundary of the intermediate region to a tip end of the blade; and

the inner region is swept forward, the intermediate region is swept rearward at a sweep angle that does not decrease, and the tip region is translated forward from the outward boundary of the intermediate region.

A blade for a gas turbine engine rotatable within a case at speeds providing supersonic flow over at least a portion of the blade, wherein the blade leading edge has a rear swept middle region ending at a tip region that is translated forward from the end of the middle region.

A blade for a gas turbine engine rotatable within a case at speeds providing supersonic flow over at least a portion of the blade, wherein the blade leading edge has a forward swept middle region ending at a tip region that is translated rearward from the end of the middle region.

5. All errors being corrected in the subject reissue application up to the time of filing this declaration, including the error identified above, arose without deceptive intent on our parts.

6. We hereby appoint John Swiatocha, Registration No. 27,955, Kenneth C. Baran, Registration No. 32682, and David M. Quinlan, Registration No. 26,641, as our attorneys to transact all business in the Patent and Trademark Office.

7. All correspondence in the above-identified application should be sent to:

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Each of us hereby declares that all statements made herein of our own knowledge are true and that all statements made on information and belief are believed to be true; and further that the statements were made with the knowledge that willful false statements and the like so made are punishable by fine or imprisonment, or both, under § 1001 of Title XVIII of United States Code, and that such willful false statements may jeopardize the validity of this application or any patent issued thereon.

Date: 6/30/99

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